US010415591B2

(12) **United States Patent**
Fessenden et al.

(10) **Patent No.: US 10,415,591 B2**
(45) **Date of Patent: Sep. 17, 2019**

(54) **GAS TURBINE ENGINE AIRFOIL**

(71) Applicant: **United Technologies Corporation**, Farmington, CT (US)

(72) Inventors: **Robert W Fessenden**, Manchester, CT (US); **Abhir A Adhate**, Hamden, CT (US); **Yuan Dong**, Glastonbury, CT (US); **Sean Nolan**, Wethersfield, CT (US); **Li Xing Pan**, Middletown, CT (US)

(73) Assignee: **UNITED TECHNOLOGIES CORPORATION**, Farmington, CT (US)

(*) Notice: Subject to any disclaimer, the term of this patent is extended or adjusted under 35 U.S.C. 154(b) by 394 days.

(21) Appl. No.: **15/272,310**

(22) Filed: **Sep. 21, 2016**

(65) **Prior Publication Data**

US 2018/0080472 A1 Mar. 22, 2018

(51) **Int. Cl.**

F04D 29/52	(2006.01)
F04D 29/32	(2006.01)
F01D 11/14	(2006.01)
F04D 29/16	(2006.01)
F01D 5/12	(2006.01)
F01D 25/24	(2006.01)

(52) **U.S. Cl.**
CPC ***F04D 29/526*** (2013.01); ***F01D 11/14*** (2013.01); ***F04D 29/164*** (2013.01); ***F04D 29/324*** (2013.01); *F01D 5/12* (2013.01); *F01D 25/24* (2013.01); *F05D 2220/32* (2013.01); *F05D 2230/50* (2013.01)

(58) **Field of Classification Search**
CPC F04D 29/526; F04D 29/164; F04D 29/324; F04D 29/642; F04D 15/0033; F01D 5/12; F01D 25/24
See application file for complete search history.

(56) **References Cited**

U.S. PATENT DOCUMENTS

4,238,170 A * 12/1980 Robideau F01D 11/08 415/173.5
4,239,452 A * 12/1980 Roberts, Jr. F01D 11/12 415/173.5

(Continued)

FOREIGN PATENT DOCUMENTS

EP 1923539 5/2008

OTHER PUBLICATIONS

Bunker, Ronald S., 2006, "Axial Turbine Blade Tips: Function, Design, and Durability", AIAA Journal of Propulsion and Power, 22, No. 2, pp. 271-285. accessed on Dec. 11, 2108 from https://arc.aiaa.org/doi/pdf/10.2514/1.11818 (Year: 2006).*

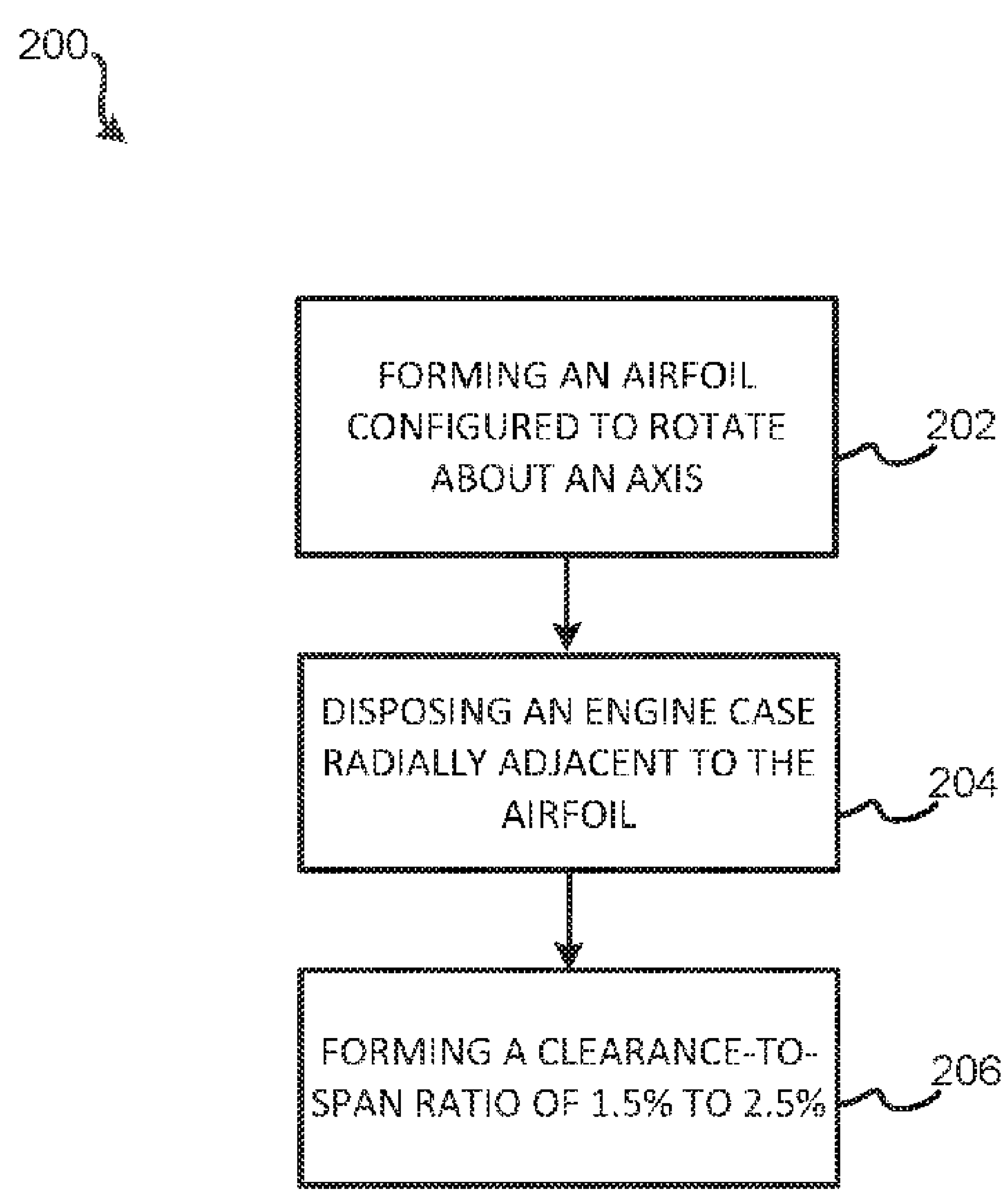
(Continued)

Primary Examiner — Richard A Edgar
Assistant Examiner — Topaz L. Elliott
(74) *Attorney, Agent, or Firm* — Snell & Wilmer, L.L.P.

(57) **ABSTRACT**

A compressor may include an airfoil configured to rotate about an axis. The airfoil may have a span measured radially from a root of the airfoil to a tip of the airfoil. A compressor case may be radially adjacent to the airfoil. The airfoil and the compressor case may define a clearance between the tip of the airfoil and a radially inner surface of the compressor case. A ratio of the clearance to the span may be between 1.5 to 2.5 percent.

13 Claims, 4 Drawing Sheets



(56) **References Cited**

U.S. PATENT DOCUMENTS

6,935,836	B2 *	8/2005	Ress, Jr.	F01D 11/025 415/173.2
7,823,389	B2 *	11/2010	Seitzer	F01D 11/24 60/782
8,562,289	B2 *	10/2013	Kobayashi	F04D 29/522 415/173.1
2006/0140756	A1	6/2006	Schwarz	
2014/0165592	A1	6/2014	Key	
2014/0216087	A1 *	8/2014	Sishtla	F04D 29/058 62/228.1
2018/0230946	A1 *	8/2018	Virtue, Jr.	F02K 3/068

OTHER PUBLICATIONS

Dong, Yang & Zheng, Xinqian & Qiushi, Li. (2014). An 11-stage axial compressor performance simulation considering the change of tip clearance in different operating conditions. Proceedings of the Institution of Mechanical Engineers, Part A: Journal of Power and Energy. 228, No. 6, pp. 614-625. (Year: 2014).*

DiOrio, Austin G. "Small Core Axial Compressor for High Efficiency Jet Aircraft" Massachusetts Institutes of Technology, 2012. accessed from https://dspace.mit.edu/bitstream/handle/1721.1/77107/825066815-MIT.pdf?sequence=2 on Dec. 12, 2018. (Year: 2012).*

Remarks filed Sep. 26, 2018 in corresponding EPO Application No. 17181892.5 (3299587) (Year: 2018).*

European Patent Office, European Search Report dated Jan. 23, 2018 in Application No. 17181892.5-1006.

* cited by examiner

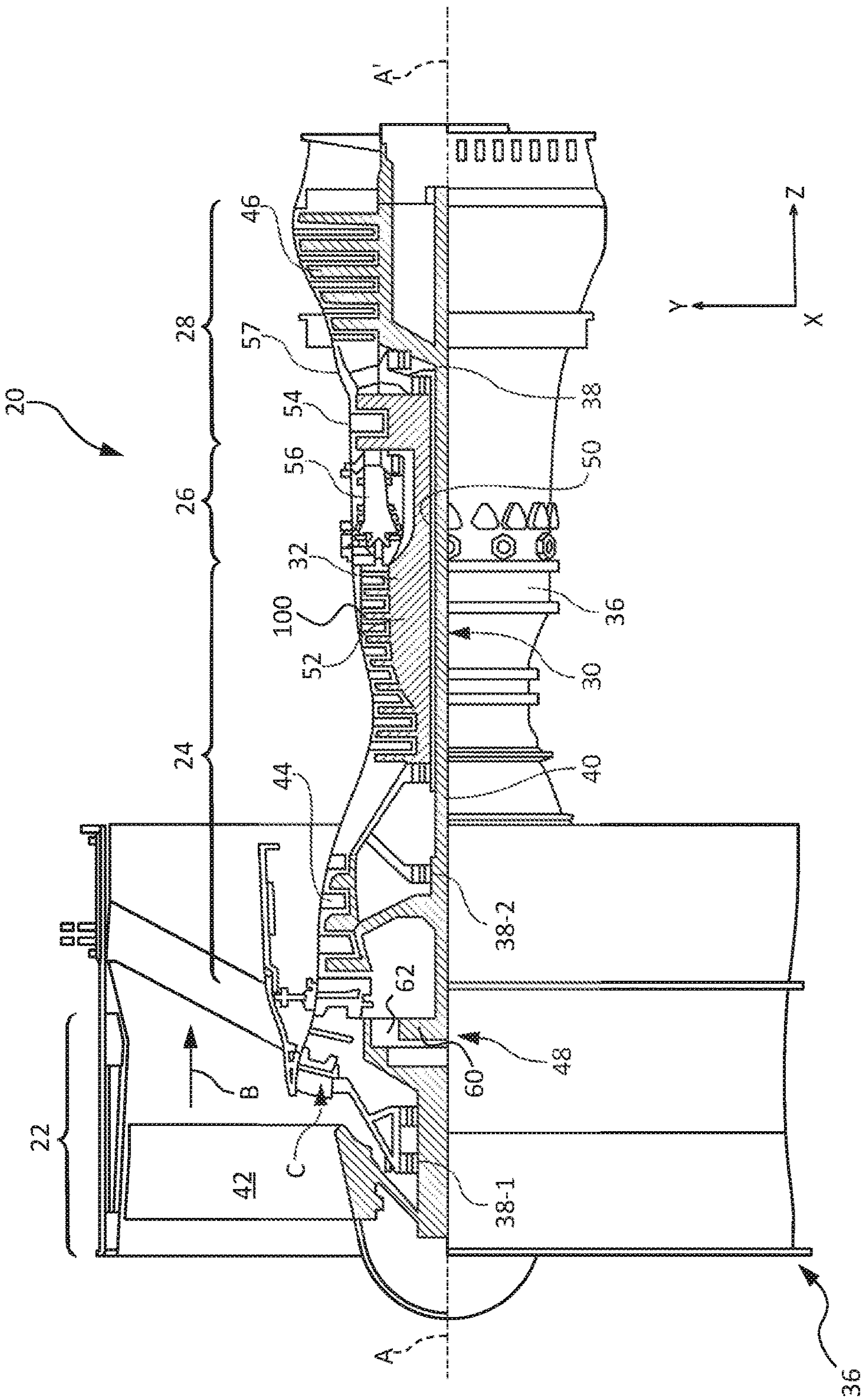

20
22
24
26
28
46
57
54
56
32
100
52
44
38
50
36
30
40
38-2
62
60
48
B
C
42
38-1
A
A'
36
X
Y
Z

FIG. 1

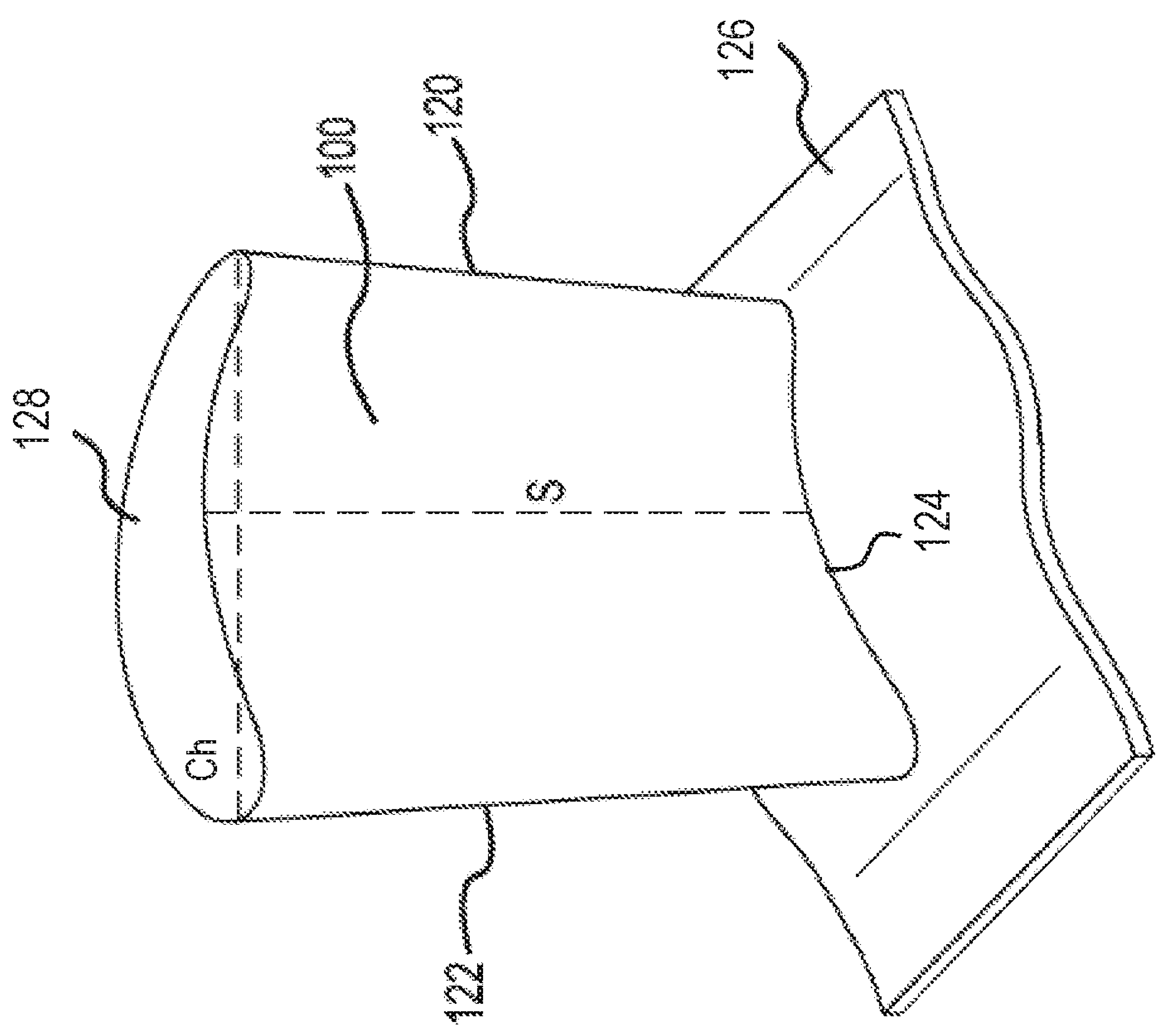


FIG. 2

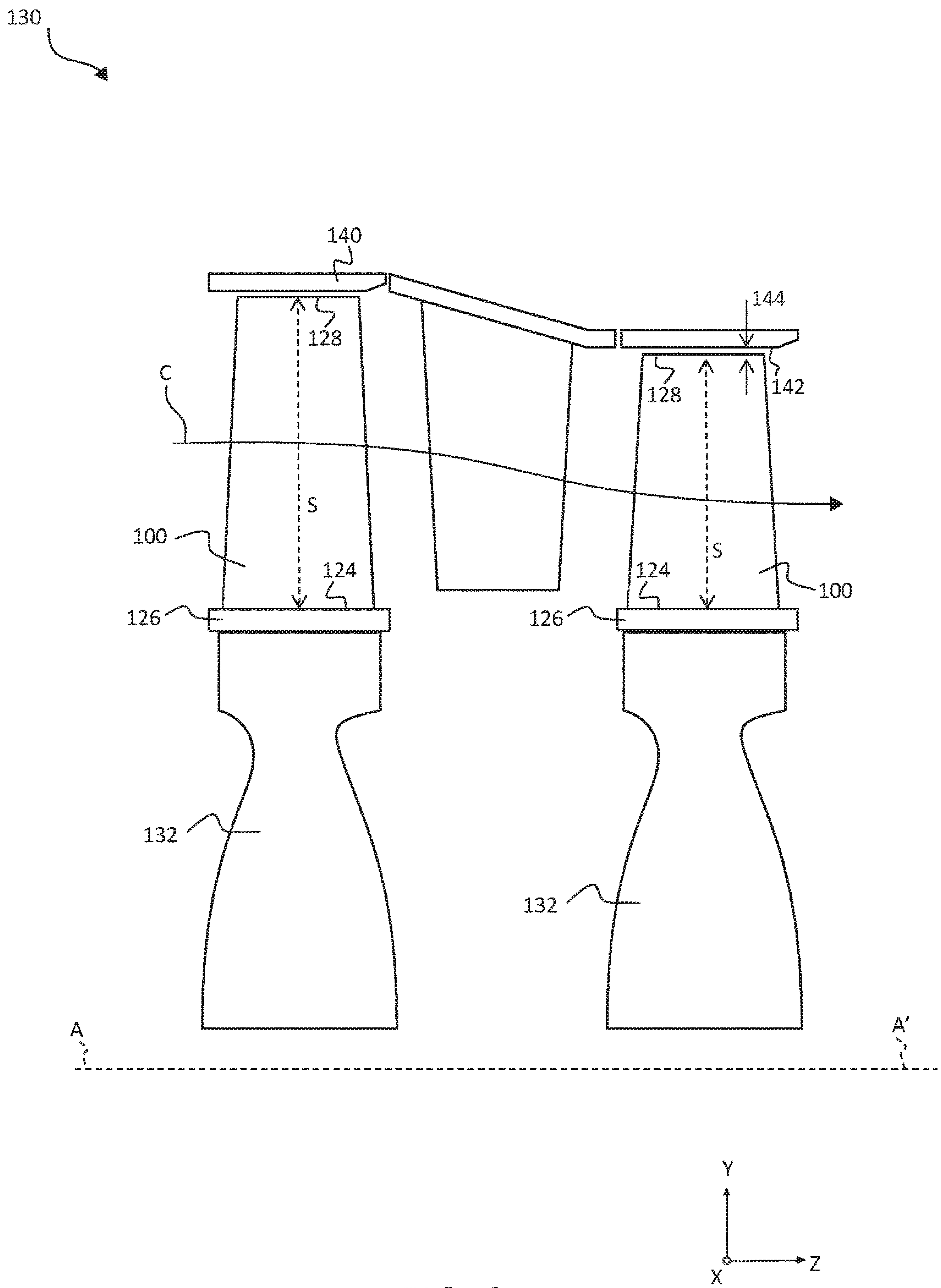

130
140
144
128
C
128
142
S
100
124
S
124
100
126
126
132
132
A
A'
Y
Z
X

FIG. 3

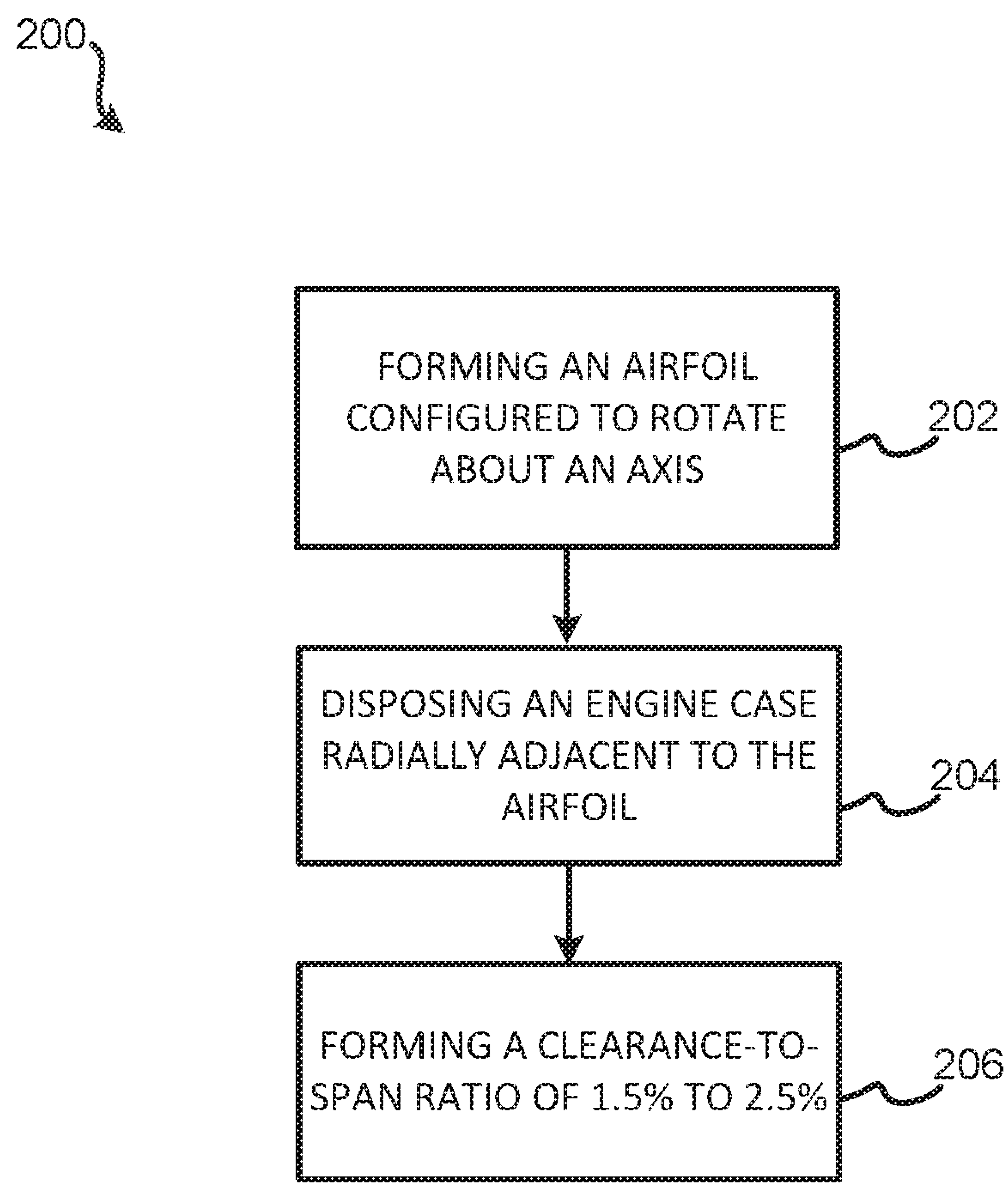

200
FORMING AN AIRFOIL CONFIGURED TO ROTATE ABOUT AN AXIS
202
DISPOSING AN ENGINE CASE RADIALLY ADJACENT TO THE AIRFOIL
204
FORMING A CLEARANCE-TO-SPAN RATIO OF 1.5% TO 2.5%
206

FIG. 4

GAS TURBINE ENGINE AIRFOIL

GOVERNMENT LICENSE RIGHTS

This disclosure was made with government support under contract No. NNC14CA36C awarded by the National Aeronautics and Space Administration (NASA). The government has certain rights in the disclosure.

FIELD

The present disclosure relates generally to components of gas turbine engines and, more specifically, to airfoils of small-core gas turbine engines.

BACKGROUND

Gas turbine engines typically include a compressor section, a combustor section and a turbine section. In general, during operation, air is pressurized in the compressor section and is mixed with fuel and burned in the combustor section to generate hot combustion gases. The hot combustion gases flow through the turbine section, which extracts energy from the hot combustion gases to power the compressor section and other gas turbine engine loads.

One potential limiting factor in gas turbine engines may be the efficiency and stability of the compression and turbine systems. Efficiency and stability of an axial compressor may be limited by the engine size and the engine operating conditions. Compressor stall may also be a limiting factor in gas turbine engines. The initiation of a stall may be driven by the tip leakage flow through the tip clearance between an airfoil and the outer diameter of the compressor. Scaling down of a gas turbine engine may result in a relatively larger tip clearance in the compressors and turbines of small-core gas turbine engines. Scaling constraints may cause small-core gas turbine engines to be less efficient relative to larger gas turbine engines.

SUMMARY

A compressor in accordance with various embodiments may comprise an airfoil configured to rotate about an axis. The airfoil may have a span measured radially from a root of the airfoil to a tip of the airfoil. A compressor case may be radially adjacent to the airfoil. The airfoil and the compressor case may define a clearance between the tip of the airfoil and a radially inner surface of the compressor case. A ratio of the clearance to the span may be between 1.5 to 2.5 percent.

In various embodiments, the ratio of the clearance to the span may be between 2.0 to 2.5 percent. The airfoil may be a high pressure compressor airfoil. A mass flow rate of the compressor may be 3.0 pounds-mass per second or less. The span of the airfoil may be between 12.7 millimeters and 15.24 millimeters. The span of the airfoil may be between 14.73 millimeters and 15.24 millimeters. The clearance may be configured to prevent contact between the tip of the airfoil and the radially inner surface of the compressor case. The airfoil may be positioned in an aft end of the compressor.

A gas turbine engine in accordance with various embodiments may comprise an engine section comprising at least one of a turbine section and a compressor section. An airfoil may be positioned within the engine section and configured to rotate about an axis. The airfoil may have a span measured radially from a root of the airfoil to a tip of the airfoil. An engine case may be radially adjacent to the airfoil. The airfoil and the engine case may define a clearance between the tip of the airfoil and a radially inner surface of the engine case. A ratio of the clearance to the span may be between 1.5 to 2.5 percent.

In various embodiments, the ratio of the clearance to the span may be between 2.0 to 2.5 percent. The airfoil may be a high pressure compressor airfoil. The engine case may be a compressor case. The span of the airfoil may be between 12.7 millimeters and 15.24 millimeters. The span of the airfoil may be between 14.73 millimeters and 15.24 millimeters. The engine section is an aft portion of a high pressure compressor.

A method of increasing gas turbine engine efficiency may comprise forming an airfoil configured to rotate about an axis. The airfoil may have a span measured radially from a root of the airfoil to a tip of the airfoil. The method may include disposing an engine case radially adjacent to the airfoil. The airfoil and the engine case may define a clearance between the tip of the airfoil and a radially inner surface of the engine case. The method may include reducing a ratio of the clearance to the span to between 1.5 to 2.5 percent.

In various embodiments, reducing the ratio of the clearance to the span includes increasing the span of the airfoil relative to a diameter of the engine case. Increasing the span of the airfoil may include increasing the span of the airfoil to between 12.7 millimeters and 15.24 millimeters. Reducing the ratio of the clearance to the span may include decreasing a diameter of the engine case. The airfoil may be a high pressure compressor airfoil and the engine case is a compressor case.

The foregoing features and elements may be combined in various combinations without exclusivity, unless expressly indicated otherwise. These features and elements as well as the operation thereof will become more apparent in light of the following description and the accompanying drawings. It should be understood, however, the following description and drawings are intended to be exemplary in nature and non-limiting.

BRIEF DESCRIPTION OF THE DRAWINGS

The subject matter of the present disclosure is particularly pointed out and distinctly claimed in the concluding portion of the specification. A more complete understanding of the present disclosure, however, may best be obtained by referring to the detailed description and claims when considered in connection with the drawing figures, wherein like numerals denote like elements.

FIG. **1** illustrates a cross-sectional view of an exemplary gas turbine engine, in accordance with various embodiments;

FIG. **2** illustrates a perspective view of an airfoil in a gas turbine engine, in accordance with various embodiments;

FIG. **3** illustrates a cross-sectional view of an engine section of a gas turbine engine, in accordance with various embodiments; and

FIG. **4** illustrates a method of increasing gas turbine engine efficiency, in accordance with various embodiments.

DETAILED DESCRIPTION

All ranges and ratio limits disclosed herein may be combined. It is to be understood that unless specifically stated otherwise, references to "a," "an," and/or "the" may include one or more than one and that reference to an item in the singular may also include the item in the plural.

The detailed description of various embodiments herein makes reference to the accompanying drawings, which show various embodiments by way of illustration. While these various embodiments are described in sufficient detail to enable those skilled in the art to practice the disclosure, it should be understood that other embodiments may be realized and that logical, chemical, and mechanical changes may be made without departing from the spirit and scope of the disclosure. Thus, the detailed description herein is presented for purposes of illustration only and not of limitation. For example, the steps recited in any of the method or process descriptions may be executed in any order and are not necessarily limited to the order presented. Furthermore, any reference to singular includes plural embodiments, and any reference to more than one component or step may include a singular embodiment or step. Also, any reference to attached, fixed, connected, or the like may include permanent, removable, temporary, partial, full, and/or any other possible attachment option. Additionally, any reference to without contact (or similar phrases) may also include reduced contact or minimal contact. Cross hatching lines may be used throughout the figures to denote different parts but not necessarily to denote the same or different materials.

As used herein, "aft" refers to the direction associated with the tail (e.g., the back end) of an aircraft, or generally, to the direction of exhaust of the gas turbine engine. As used herein, "forward" refers to the direction associated with the nose (e.g., the front end) of an aircraft, or generally, to the direction of flight or motion.

As used herein, "distal" refers to the direction radially outward, or generally, away from the axis of rotation of a turbine engine. As used herein, "proximal" refers to a direction radially inward, or generally, towards the axis of rotation of a turbine engine.

With reference to FIG. **1**, a gas-turbine engine **20** is provided. Gas-turbine engine **20** may be a two-spool turbofan that generally incorporates a fan section **22**, a compressor section **24**, a combustor section **26** and a turbine section **28**. Alternative engines may include, for example, an augmentor section among other systems or features. In operation, fan section **22** can drive coolant along a bypass flow-path B while compressor section **24** can drive coolant along a path of core airflow C for compression and communication into combustor section **26** then expansion through turbine section **28**. Although depicted as a turbofan gas-turbine engine **20** herein, it should be understood that the concepts described herein are not limited to use with turbofans as the teachings may be applied to other types of turbine engines including three-spool architectures.

Gas-turbine engine **20** may generally comprise a low speed spool **30** and a high speed spool **32** mounted for rotation about an engine central longitudinal axis A-A' relative to an engine static structure or engine case **36** via several bearing systems **38**, **38-1**, and **38-2**. It should be understood that various bearing systems **38** at various locations may alternatively or additionally be provided, including for example, bearing system **38**, bearing system **38-1**, and bearing system **38-2**.

Low speed spool **30** may generally comprise an inner shaft **40** that interconnects a fan **42**, a low pressure compressor section **44** and a low pressure turbine section **46**. Inner shaft **40** may be connected to fan **42** through a geared architecture **48** that can drive fan **42** at a lower speed than low speed spool **30**. Geared architecture **48** may comprise a gear assembly **60** enclosed within a gear housing **62**. Gear assembly **60** couples inner shaft **40** to a rotating fan structure. High speed spool **32** may comprise an outer shaft **50** that interconnects a high pressure compressor **52** and high pressure turbine **54**. A combustor **56** may be located between high pressure compressor **52** and high pressure turbine **54**. Mid-turbine frame **57** may support one or more bearing systems **38** in turbine section **28**. Inner shaft **40** and outer shaft **50** may be concentric and rotate via bearing systems **38** about the engine central longitudinal axis A-A', which is collinear with their longitudinal axes. As used herein, a "high pressure" compressor or turbine experiences a higher pressure than a corresponding "low pressure" compressor or turbine.

The core airflow C may be compressed by low pressure compressor section **44** then high pressure compressor **52**, mixed and burned with fuel in combustor **56**, then expanded over high pressure turbine **54** and low pressure turbine **46**. Turbines **46**, **54** rotationally drive the respective low speed spool **30** and high speed spool **32** in response to the expansion.

Gas-turbine engine **20** may be, for example, a high-bypass ratio geared aircraft engine. In various embodiments, the bypass ratio of gas-turbine engine **20** may be greater than about six (6). In various embodiments, the bypass ratio of gas-turbine engine **20** may be greater than ten (10). In various embodiments, geared architecture **48** may be an epicyclic gear train, such as a star gear system (sun gear in meshing engagement with a plurality of star gears supported by a carrier and in meshing engagement with a ring gear) or other gear system. Geared architecture **48** may have a gear reduction ratio of greater than about 2.3 and low pressure turbine **46** may have a pressure ratio that is greater than about five (5). In various embodiments, the bypass ratio of gas-turbine engine **20** is greater than about ten (10:1). In various embodiments, the diameter of fan **42** may be significantly larger than that of the low pressure compressor section **44**, and the low pressure turbine **46** may have a pressure ratio that is greater than about five (5:1). Low pressure turbine **46** pressure ratio may be measured prior to inlet of low pressure turbine **46** as related to the pressure at the outlet of low pressure turbine **46** prior to an exhaust nozzle. It should be understood, however, that the above parameters are exemplary of various embodiments of a suitable geared architecture engine and that the present disclosure contemplates other turbine engines including direct drive turbofans.

Each of the compressor section **24** and the turbine section **28** may include alternating rows of rotor blade assemblies and vane assemblies (shown schematically) that carry airfoils that extend into the path of core airflow C. The blade assemblies have one or more sets of rotating blades, which may rotate about engine central longitudinal axis A-A'. Blade assemblies may rotate with respect to static components of gas turbine engine **20**, such as the vane assemblies and engine case **36**. A clearance is maintained between rotating blade assemblies and static components in order to allow blade assemblies to rotate relative to the static components without contacting the static components. For example, compressor section **24** includes airfoils **100**, blades and/or vanes, in the path of core airflow C. Airfoils **100** may rotate with respect to engine case **36**, which may be a compressor case. Sufficient clearance is maintained to allow for vibration and thermal expansion of these components within gas turbine engine **20**, while avoiding airfoil **100** contact with engine case **36**. Regardless of engine size, a minimum tip clearance is needed between rotating airfoils **100** and engine case **36** to maintain function (i.e. by preventing contact between airfoils **100** and engine case **36**) during various operating conditions.

The present disclosure relates to methods of improving efficiency in compressor section **24** and the turbine section **28**, particularly in small-core gas turbine engines, wherein "small-core" refers to a core size with a mass flow rate of 3.0 pounds-mass per second (lbm/s) or less. Small-core gas turbine engines may include scaled down engine components relative to larger gas turbine engines. In small-core gas turbine engines, the compressor section **24**, for example, may include a scaled down size of the compressor case and airfoils **100** relative to a larger gas turbine engine. A tip clearance may not be scalable with the compressor components. In order to maintain sufficient tip clearance, airfoils **100** may not be scaled down proportionally with the engine case **36**. For example, in scaling a compressor section **24** down by a factor of two, the tip clearance may not also be reduced by the same factor without exceeding the minimum tip clearance and causing airfoil **100** contact with the and engine case **36**. Thus, a tip clearance may not be proportionally scalable to a small-core engine. The same tip clearance for a larger compressor section **24** may be functional (i.e. by preventing contact between airfoils **100** and engine case **36**) in a smaller compressor section **24**, but may lead to more inefficiencies in a smaller compressor section **24** relative to a larger compressor section **24**. For example, a 0.254 millimeter (mm) (0.01 inch) tip clearance may impact efficiency in a smaller compressor section **24** more than in a larger compressor section **24**. High pressure compressor **52** may have a smaller diameter and smaller span airfoils compared low pressure compressor **44**. Similarly, an aft section of a compressor, such as low pressure compressor **44** and/or high pressure compressor **52**, may have a smaller diameter and smaller span airfoils compared to a forward end of the same compressor. Tip clearance may have a greater impact on efficiency in the high pressure compressor **52** than in the low pressure compressor **44**, and further may have a greater impact on efficiency in an aft section of high pressure compressor **52** than in a forward section. Tip clearance may particularly impact the aft-most section of high pressure compressor **52**, for example, at an exit of high pressure compressor **52**.

As discussed herein, a clearance-to-span ratio may be designed to reduce the impact of tip clearance on efficiency in small-core gas turbine engines. In various embodiments, the blade assemblies of gas turbine engine **20** have a reduced clearance-to-span ratio for improved stability and efficiency.

With reference to FIG. **2**, an airfoil **100** is shown in accordance with various embodiments. Airfoil **100** comprises trailing edge **120** facing an aft direction in a gas turbine engine and leading edge **122** facing a forward direction in the gas turbine engine. Airfoil **100** may include a hub end or root **124** at a radially inner edge of airfoil **100**. Root **124** of airfoil **100** may be coupled to a platform **126**. For example, airfoil **100** may be coupled and secured to platform **126** by welding, machining, press fitting, and any other acceptable method of coupling. Platform **126** may form a radially inner boundary of a flow path of core airflow C in the gas turbine engine. A radially outer edge or tip **128** of airfoil **100** may be located radially outward from the root **124**. Tip **128** of airfoil **100** may face radially outward when airfoil **100** is installed in a rotating compressor section of a gas turbine engine. Airfoil **100** may further include a generally concave pressure side and a generally convex suction side joined together at the respective trailing edge **120** and leading edge **122**. A span S of airfoil **100** extends between the root **124** and tip **128** of the airfoil **100**. A span S of the airfoil may be described as the height or length of the airfoil in the radial direction and measured radially from root **124** to tip **128** of airfoil **100**. A chord Ch of the airfoil **100** extends between the leading edge **122** and the trailing edge **120**. A chord at top **124** of airfoil **100** (i.e., the tip chord) is illustrated as chord Ch.

With reference to FIG. **3**, a portion of an engine section **130** is shown in accordance with various embodiments. Engine section **130** may be a compressor section **24** or a turbine section **28**. In various embodiments, engine section **130** may be a small-core axial-flow compressor and/or another high pressure compressor or low pressure compressor. Engine section **130** may be a compressor having a mass flow rate of 3.0 lbm/s or less. Engine section **130** may be an aft portion of a high pressure compressor **52** (from FIG. **1**), and more particularly, may be an exit stage or aft-most stage of high pressure compressor **52**. Engine section **130** may include a plurality of airfoils **100** each coupled to a disk **132** by platform **126**. Disk **132** may be centered on the rotation axis of the gas turbine engine with a plurality of airfoils **100** attached to the disk **132** and spaced apart in the circumferential or tangential direction. Disk **132** with airfoils **100** may be configured to rotate about engine central longitudinal axis A-A'. Thus, airfoil **100** may be a rotating member of engine section **130**.

Airfoil **100** may rotate with respect to a static member or engine case **140** of engine section **130**. Engine case **140** may have a cylindrical wall defining a radially outer boundary of a flow path of core airflow C. Engine case **140** may be, for example, a compressor case. Engine case **140** may extend circumferentially and radially surround airfoils **100**. Engine case **140** may include an inner surface **142** adjacent to the tips **128** of the airfoils **100**. A radial distance between this inner surface **142** of engine case **140** and the tip **128** of airfoil **100** defines a tip clearance, illustrated by clearance **144**. Thus, engine case **140** and airfoil **100** may define a clearance **144** therebetween. Clearance **144** may be a gap between tip **128** of airfoil **100** and inner surface **142** of engine case **140** that allows airfoil **100** to rotate within engine section **130** without contacting engine case **140**. A minimum distance of clearance **144** is maintained to prevent contact between tip **128** of airfoil **100** and inner surface **142** of engine case **140**.

In various embodiments, a ratio of clearance **144** to span S of airfoils **100** ("clearance-to-span ratio") correlates to an efficiency of the engine section **130**. The mass flow rate of core airflow C correlates to the velocity of the airflow and the cross-sectional area of the flowpath, which is determined in part by the span S of airfoil **100**. Clearance **144** may allow air to escape from core airflow C through the clearance **144**. Reducing a distance of clearance **144** reduces the amount of core airflow C escaping through clearance **144**, and thereby increases the stability and efficiency of the engine section **130**. However as discussed above, a minimum distance for clearance **144** is maintained to prevent contact between airfoil **100** and engine case **140**. Typically, a clearance **144** is translated as a fixed value from a larger sized engine to a smaller sized engine without scaling the clearance **144**, i.e., the same clearance is used for a larger and a smaller sized engine, because proportionally scaling clearance **144** down to a small-core engine would exceed the minimum tip clearance and result in contact between rotating and static structures. For a fixed clearance, the ratio of clearance **144** to span S in a smaller engine is relatively greater than in a larger engine. Similarly for the fixed clearance, the ratio of clearance **144** to span S in smaller engine section **130** is relatively greater than in a larger engine section **130**. Also for a fixed clearance, a span S of airfoils **100** would not be proportionately scaled down. In other words, when scaling down an engine while maintaining the fixed clearance, a span S would be made disproportionately smaller in the smaller engine to maintain the fixed clearance. Stated another way, the span of an airfoil in a smaller engine (or engine section) is proportionally smaller relative to it's engine components as compared to the relative span of an airfoil in a larger engine (or engine section) with the same clearance.

In various embodiments to improve efficiency of a smaller engine, a clearance-to-span ratio in engine section **130** may be reduced relative to an engine section configured with the fixed clearance of a larger engine. For example, rather than fixing the clearance or proportionally scaling the clearance, airfoils **100** and clearance **144** may be configured to optimize the clearance-to-span ratio in engine section **130**. Clearance-to-span ratio of an airfoil **100** may be reduced by increasing span S and/or by decreasing clearance **144**. For example, clearance **144** may be designed with a smaller distance in a smaller engine section **130**, and still remain within a minimum tip clearance for preventing contact between airfoils **100** and engine case **140**. Clearance **144** may be reduced in engine section **130** by decreasing a diameter of engine case **140** relative to the span S of airfoil **100** or by increasing the span S of airfoil **100** relative to a diameter of engine case **140**. Clearance **144** may be reduced by both decreasing a diameter of engine case **140** and by increasing the span S of airfoil **100**. The span S of airfoil **100** may be increased by extending airfoil **100** radially inward (negative y direction) at root **124** and/or radially outward (positive y direction) at tip **128**. For example, lengthening airfoil **100** at root **124** may include decreasing a diameter of disk **132**. Decreasing a diameter of disk **132** allows an airfoil **100** having a larger span S to fit within engine case **140**, while maintaining clearance **144**, and thereby decreasing the clearance-to-span ratio. Thus, span S of airfoil **100** may be increased with or without decreasing clearance **144** and with or without changing a diameter of engine case **140**.

In various embodiments, the span S of airfoil **100** is increased relative to clearance **144** by between 20 to 30 percent, and more specifically between 20 to 25 percent. For example, in a small-core compressor, the span S of airfoil **100** may be between about 12.7 millimeters (mm) (0.5 inches) to 15.24 mm (0.6 inches), wherein "about" in this context only means+/−0.25 mm (0.01 inches). More specifically, the span S of airfoil **100** may be between about 14.73 mm (0.58 inches) to 15.24 mm (0.6 inches), wherein "about" in this context only means+/−0.25 mm (0.01 inches). A clearance-to-span ratio for engine section **130** may be described as a ratio of clearance **144** to span S. Increasing the span S of airfoil by 20 to 30 percent may reduce the clearance-to-span ratio to between 0.015:1 and 0.025:1, which may also be expressed as 1.5% to 2.5%. More specifically, the clearance-to-span ratio may be between 0.020:1 and 0.025:1, which may also be expressed as 2.0% to 2.5%. A clearance-to-span ratio of 1.5% to 2.5% may increase the polytropic efficiency of engine section **130** by 1%. A stall margin may be impacted by an increased span S. Thus, airfoils **100** may further be configured to offset stall margin losses by configuring an angle of attack of airfoils **100** to reduce risk of misalignment of airfoils **100** with core airflow C.

In various embodiments, airfoil **100** may be a blade of a high pressure compressor, and more particularly, may be an aft-most blade of a high pressure compressor, such as a small-core high pressure compressor. A compressor diffuses core airflow C to increase the pressure of core airflow C. Diffusion may be expressed a ratio of the exit velocity to the inlet velocity of core airflow C through the compressor. Reducing a clearance-to-span ratio by increasing the span S of airfoil **100** may increase compressor diffusion. Increasing span S may include extending airfoil **100** in a radially outward or radially inward direction. Reducing a diameter of disk **132** may allow airfoil **100** to be configured with a larger span S. Reducing a clearance-to-span ratio may further include decreasing clearance **144** and/or decreasing a diameter of the compressor case to increase compressor diffusion. The span S of airfoil **100** in a small-core compressor may be configured to increase compressor diffusion as compared to typical small-core compressors. A geometrical shape and angle of attack of airfoil **100** may be configured to mitigate stall margin losses.

With reference to FIG. **4**, a method **200** for increasing gas turbine engine efficiency is shown in accordance with various embodiments. Method **200** may comprise the steps of forming an airfoil configured to rotate about an axis (step **202**) disposing an engine case radially adjacent to the airfoil (step **204**), and forming a ratio of the clearance to the span in a range of 1.5 to 2.5 (step **206**). The airfoil **100** has a span S measured radially from a root **124** of the airfoil **100** to a tip **128** of the airfoil **100**. Step **206** may further comprise increasing the span S of the airfoil **100** relative to a diameter of the engine section **130**. Increasing the span S of the airfoil **100** may include decreasing a diameter of disk **132** to extend airfoil **100** radially inward. Step **206** may further comprise reducing the ratio of the clearance **144** to the span S by increasing the span S of the airfoil **100** relative to a diameter of the engine section **130**. Step **206** may further comprise increasing the span S of the airfoil **100** to between 12.7 millimeters and 15.24 millimeters. Step **206** may further comprise reducing the ratio of the clearance **144** to the span S by decreasing a diameter of the engine case **140** relative to the span S of the airfoil **100**. The airfoil **100** may be a high pressure compressor airfoil and engine case **140** may be a compressor case.

Benefits and other advantages have been described herein with regard to specific embodiments. Furthermore, the connecting lines shown in the various figures contained herein are intended to represent exemplary functional relationships and/or physical couplings between the various elements. It should be noted that many alternative or additional functional relationships or physical connections may be present in a practical system. However, the benefits, advantages, and any elements that may cause any benefit or advantage to occur or become more pronounced are not to be construed as critical, required, or essential features or elements of the disclosure. The scope of the disclosure is accordingly to be limited by nothing other than the appended claims, in which reference to an element in the singular is not intended to mean "one and only one" unless explicitly so stated, but rather "one or more." Moreover, where a phrase similar to "at least one of A, B, or C" is used in the claims, it is intended that the phrase be interpreted to mean that A alone may be present in an embodiment, B alone may be present in an embodiment, C alone may be present in an embodiment, or that any combination of the elements A, B and C may be present in a single embodiment; for example, A and B, A and C, B and C, or A and B and C.

Systems, methods and apparatus are provided herein. In the detailed description herein, references to "various embodiments", "one embodiment", "an embodiment", "an example embodiment", etc., indicate that the embodiment described may include a particular feature, structure, or characteristic, but every embodiment may not necessarily include the particular feature, structure, or characteristic.

Moreover, such phrases are not necessarily referring to the same embodiment. Further, when a particular feature, structure, or characteristic is described in connection with an embodiment, it is submitted that it is within the knowledge of one skilled in the art to affect such feature, structure, or characteristic in connection with other embodiments whether or not explicitly described. After reading the description, it will be apparent to one skilled in the relevant art(s) how to implement the disclosure in alternative embodiments.

Furthermore, no element, component, or method step in the present disclosure is intended to be dedicated to the public regardless of whether the element, component, or method step is explicitly recited in the claims. No claim element is intended to invoke 35 U.S.C. 112(f) unless the element is expressly recited using the phrase "means for." As used herein, the terms "comprises", "comprising", or any other variation thereof, are intended to cover a non-exclusive inclusion, such that a process, method, article, or apparatus that comprises a list of elements does not include only those elements but may include other elements not expressly listed or inherent to such process, method, article, or apparatus.

What is claimed is:

1. A compressor, comprising:

an airfoil configured to rotate about an axis, wherein the airfoil has a span measured radially from a root of the airfoil to a tip of the airfoil; and

a compressor case radially adjacent to the airfoil, wherein the airfoil and the compressor case define a clearance between the tip of the airfoil and a radially inner surface of the compressor case and wherein a ratio of the clearance to the span is between 1.5 and 2.5 percent, wherein a mass flow rate of the compressor is 3.0 pounds-mass per second or less, wherein the span of the airfoil is between about 14.73 millimeters and about 15.24 millimeters.

2. The compressor of claim **1**, wherein the ratio of the clearance to the span is between 2.0 and 2.5 percent.

3. The compressor of claim **1**, wherein the airfoil is a high pressure compressor airfoil.

4. The compressor of claim **1**, wherein the clearance is configured to prevent contact between the tip of the airfoil and the radially inner surface of the compressor case.

5. The compressor of claim **1**, wherein the airfoil is positioned in an aft end of the compressor.

6. A gas turbine engine, comprising:

an engine section comprising a compressor;

an airfoil positioned within the engine section and configured to rotate about an axis, wherein the airfoil has a span measured radially from a root of the airfoil to a tip of the airfoil; and

an engine case radially adjacent to the airfoil, wherein the airfoil and the engine case define a clearance between the tip of the airfoil and a radially inner surface of the engine case and wherein a ratio of the clearance to the span is between 1.5 and 2.5 percent, wherein a mass flow rate of the compressor is 3.0 pounds-mass per second or less, wherein the span of the airfoil is between about 14.73 millimeters and about 15.24 millimeters.

7. The gas turbine engine of claim **6**, wherein the ratio of the clearance to the span is between 2.0 and 2.5 percent.

8. The gas turbine engine of claim **6**, wherein the airfoil is a high pressure compressor airfoil.

9. The gas turbine engine of claim **8**, wherein the engine case is a compressor case.

10. The gas turbine engine of claim **6**, wherein the engine section is an aft portion of a high pressure compressor.

11. A method of increasing gas turbine engine efficiency, comprising:

forming an airfoil configured to rotate about an axis, wherein the airfoil has a span measured radially from a root of the airfoil to a tip of the airfoil, wherein the airfoil is a high pressure compressor airfoil having a mass flow rate of 3.0 pounds-mass per second or less;

disposing an engine case radially adjacent to the airfoil, wherein the airfoil and the engine case define a clearance between the tip of the airfoil and a radially inner surface of the engine case; and

forming a ratio of the clearance to the span in a range of 1.5 to 2.5 percent, wherein the forming the ratio of the clearance to the span includes increasing the span of the airfoil to between about 14.73 millimeters and about 15.24 millimeters.

12. The method of claim **11**, wherein the forming the ratio of the clearance to the span includes decreasing a diameter of the engine case relative to the span of the airfoil.

13. The method of claim **11**, wherein the engine case is a compressor case.

* * * * *